

Goddard Space Flight Center

# **Electrical Power System Architectures For In-House NASA/GSFC Missions**

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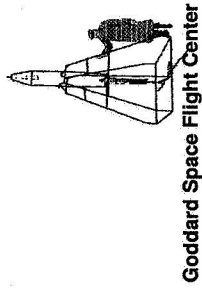
Greenbelt, MD

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# Acknowledgements

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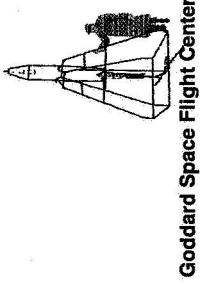


- Amri Hernandez-Pellerano
- Denney Keys
- Thomas Spitzer
- Karen Stewart

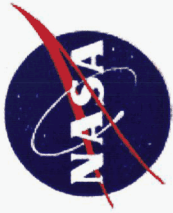


# Agenda

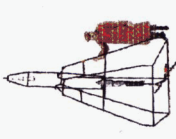
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- EPS Designs for:
  - Space Technology 5 (ST-5) Mission
  - Solar Dynamics Observatory (SDO) Mission
  - Lunar Reconnaissance Orbiter (LRO) Mission
- History
- Advancement of PSE
- Summary

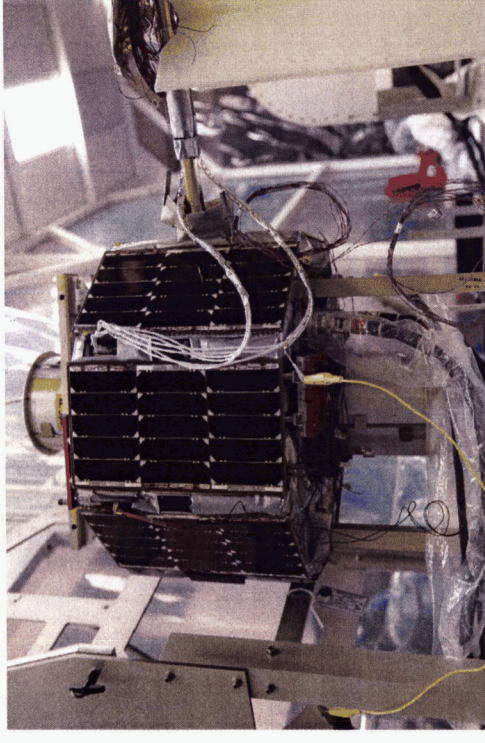


# ST-5 Mission – Mission Parameters



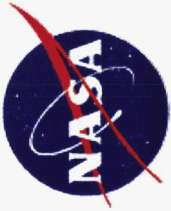
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- Orbit: in a near-Earth polar elliptical from  
~ 300 km to 4,500 km
  - 40-200 km apart from each other
- Launch Date: March 22, 2006
- Launched on: Pegasus XL Rocket
- EPS system designed for 12-21 W of load power
- Mission Design Life: 3 months
- Main mission goals:
  - Testing new micro-spacecraft technologies and operation techniques. The three spacecraft will perform coordinated multi-point measurements of the Earth's magnetic field.
- New Technologies:
  - Moved Li-Ion Battery from a TRL level 4 to a 8 (9 after launch)
  - Highest efficiency solar cells at time of contract award (Research cells)



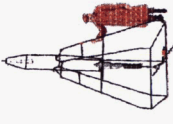
- 3 micro-satellites – Each:
- Weighs approximately 25 kgs when fully fueled
  - Is about the size of a 19" tv



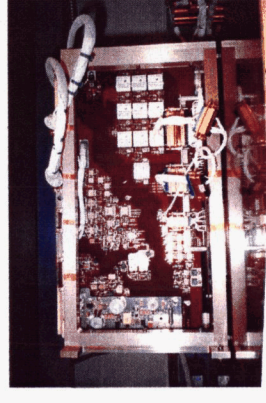
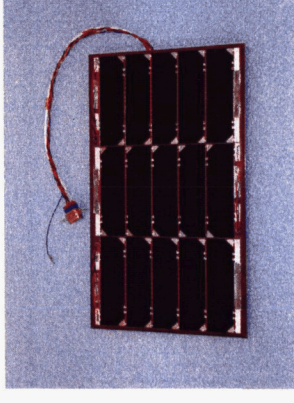


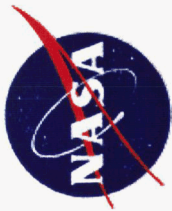
# ST-5 EPS Overview

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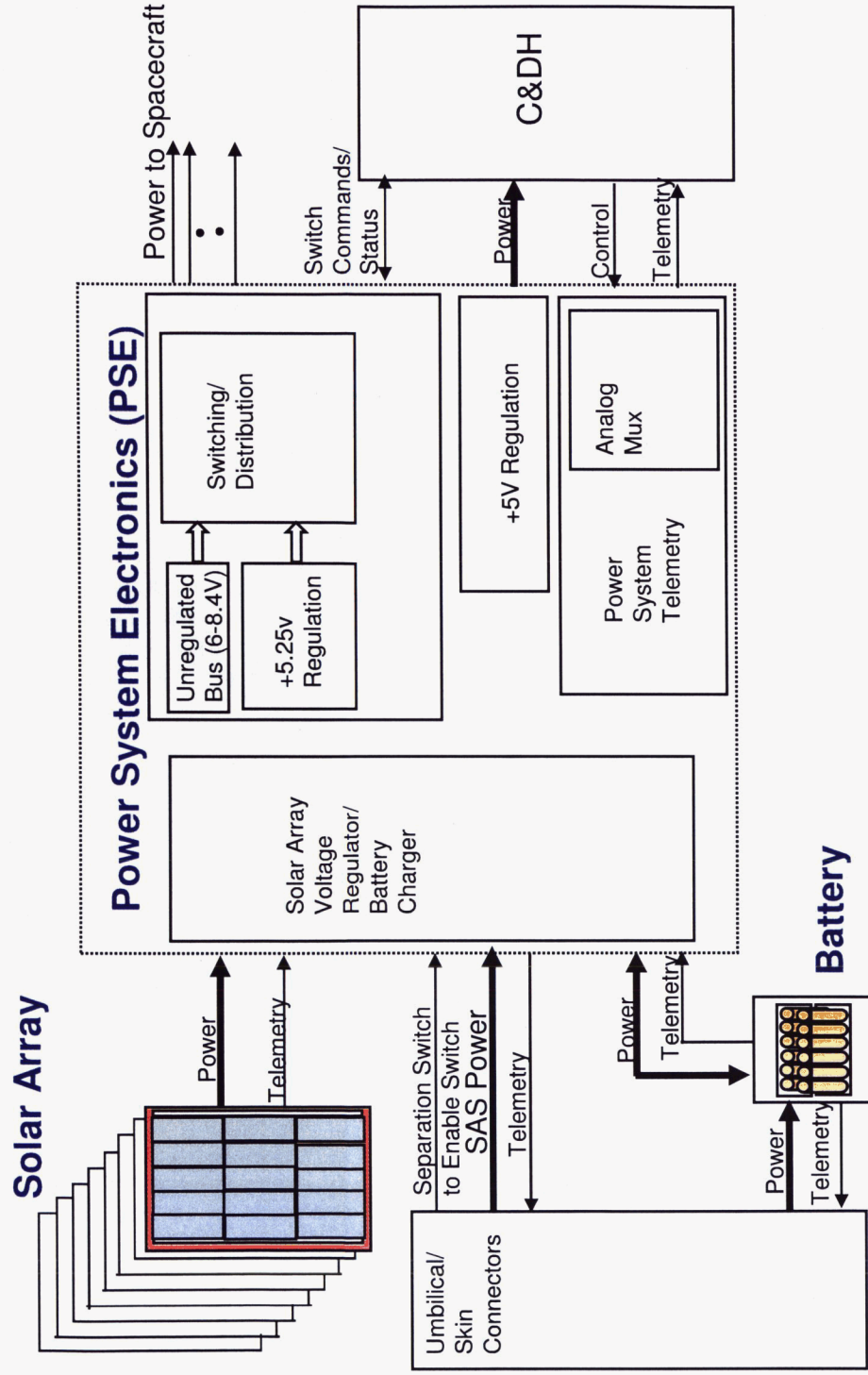
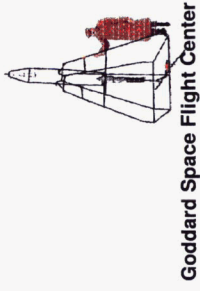


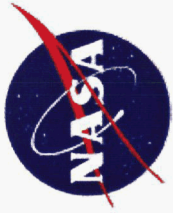
- Battery
  - 7.5 AHr Li-Ion battery made up of Sony 1.5 AHr cells
    - 2 in series, 6 strings in parallel
  - Built and tested by AEA Technology Space, UK
- Solar Array
  - 8 identical panels
  - Utilizes 28.05% TJ GaAs cells
  - Designed to provide 27W at BOL, 12.1W at EOL
  - Built and tested by Emcore, USA
- PSE:
  - DET architecture with battery connected directly to the electrical bus
  - Provides unregulated 6-8.4V power to spacecraft
  - Provides regulated 5.0V and 5.25V power to spacecraft
  - Provides switched and unswitched services for spacecraft loads





# ST-5 EPS Block Diagram

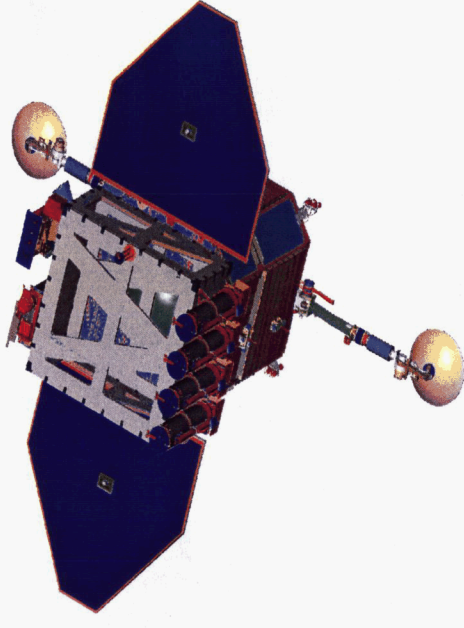




# SDO – Mission Parameters

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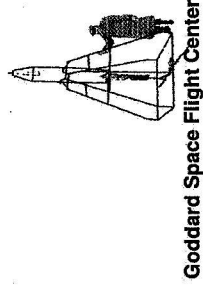
- Orbit: GEO - Sync. Orbit, Inclination 28.5 degrees
- Launch Date: April 2008
- Launched on: Atlas V
- EPS system designed for 1450 W of load power
  - 1500 Watts peak power
- Mission Design Life: 5 years
- 3 Instruments
- Main mission goals:
  - SDO is the 1st mission under the Living With a Star (LWS) Program.
  - Primary goal of the SDO mission is to understand the nature and source of the solar variations that affect life and society by determining: How and why does the sun vary?; How does the Earth respond?; And how does it impact humanity?
- New Technology:
  - Li-Ion battery





# SDO EPS Overview

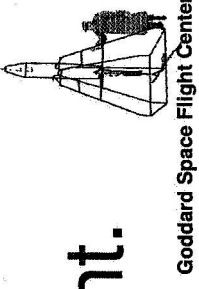
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- Battery
  - 120 Ah Li-Ion battery made up of Sony 1.5 Ahr cells
    - 8 in series, 104 strings in parallel
  - Built and tested by ABSL Power Solutions, UK
- Solar Array
  - 2 Fixed identical wings
  - Utilizes 28.5% efficient solar cells
  - Designed to provide 1068 W at BOL, 933 W at EOL
  - Built and tested by SpectroLab
- PSE
  - DET architecture with battery connected directly to the electrical bus
  - Provides unregulated bus between 22-35V
  - Single fault tolerant architecture
  - cPCI Interface



# SDO EPS Overview – PSE cont.

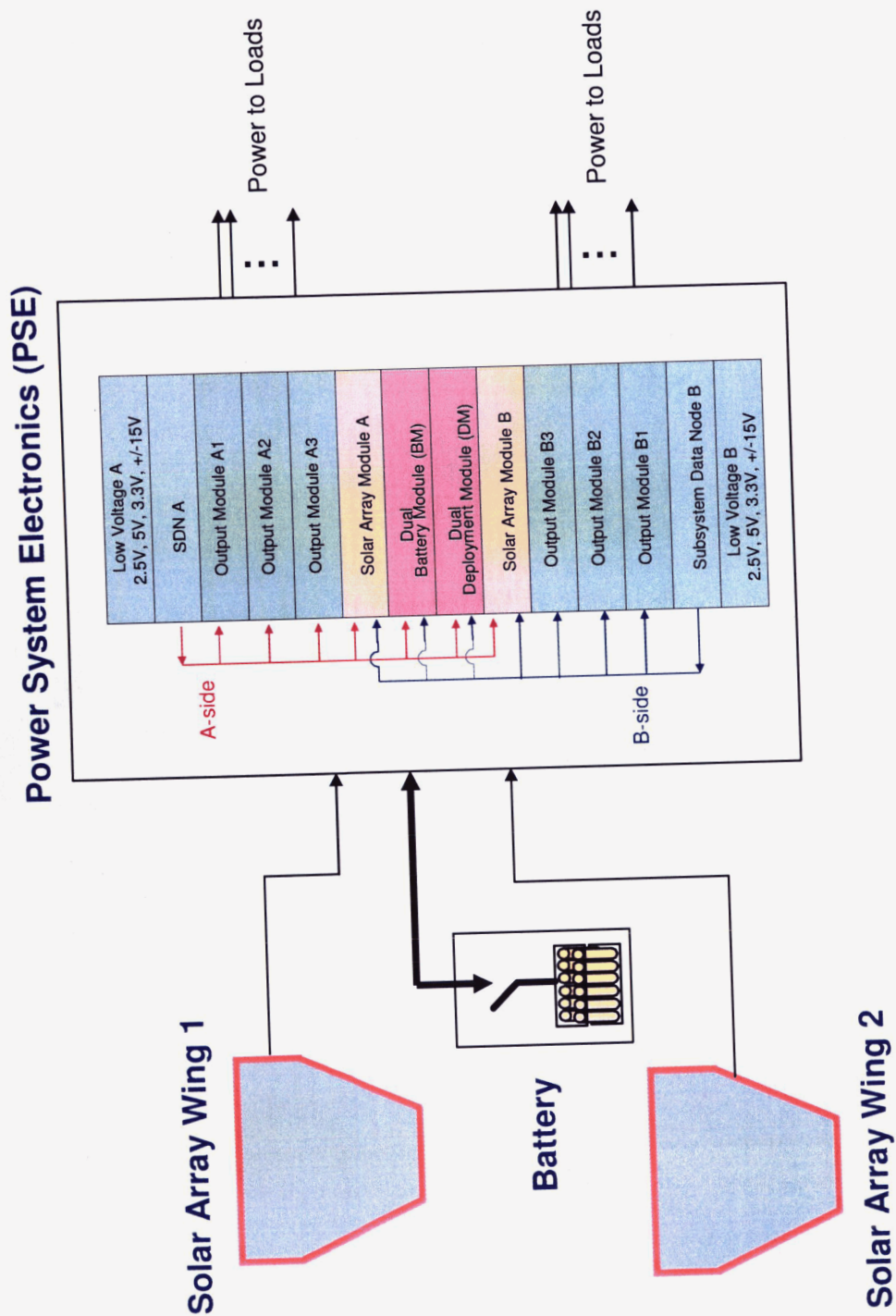
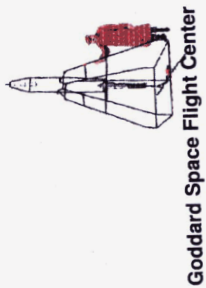


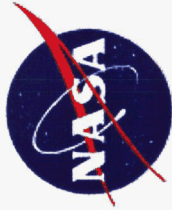
- Battery Charge Regulation is a combination of hardware and software.
- Process or shunt solar array power via PWM Boost converter
- Power Distribution
  - Provide switched or un-switched power feeds to all spacecraft loads
  - Provide over-current protection on all feeds
- Deployment Functions
  - Sense launch vehicle separation signals and provide autonomous, hardware controlled deployment of the Solar Array
- Autonomous Fault Detection and Correction (Back-Up to Main FDCs):
  - Excessive discharge → load shed
  - If discharge continues battery will be disconnected from bus until ground commanded to be reconnected (system will operate in OV control mode)
  - Battery OT protection
  - Battery OV protection



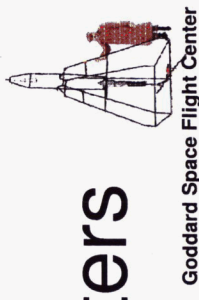


# SDO EPS Block Diagram





# LRO Mission – Mission Parameters



- Orbit: Lunar Orbit
  - 50 km +/- 20 km above mean lunar surface
  - Orbital Inclination 90 degrees +/- 1 degree
- Launch Date: October 31, 2008
- Launched on: Delta IV or Atlas V
- EPS system designed for 800 W of Load Power
- Mission Design Life: 14 months
- 7 Instruments
- Main mission goals:
  - Conduct investigations that will be specifically targeted to prepare for and support future human exploration of the moon
- NEW Technology:
  - Li-Ion battery
- Key Design Drivers that Impacted EPS:
  - Beta angle variation from 0 to 90 degrees
  - Eclipse variation from 0 to 48 minutes

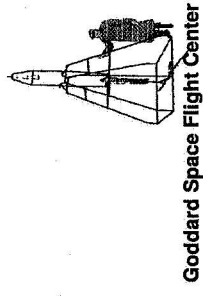






# LRO EPS Overview

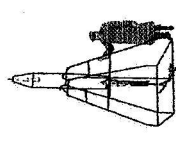
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- Battery
  - 80 Ah Li-Ion Battery
  - Cell balancing and cell by-pass required in RFP
  - Battery Vendor not yet selected
- Solar Array
  - 77 Modules
  - 1 wing, 3 panels
  - 10 m<sup>2</sup> total area
  - 2 axis articulated
  - -170 C to +145C
  - GSFC will fabricate panel/wing
  - Solar Cell vendor not yet selected

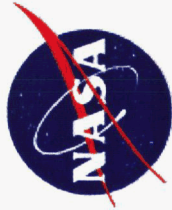


# LRO EPS Overview – PSE cont.

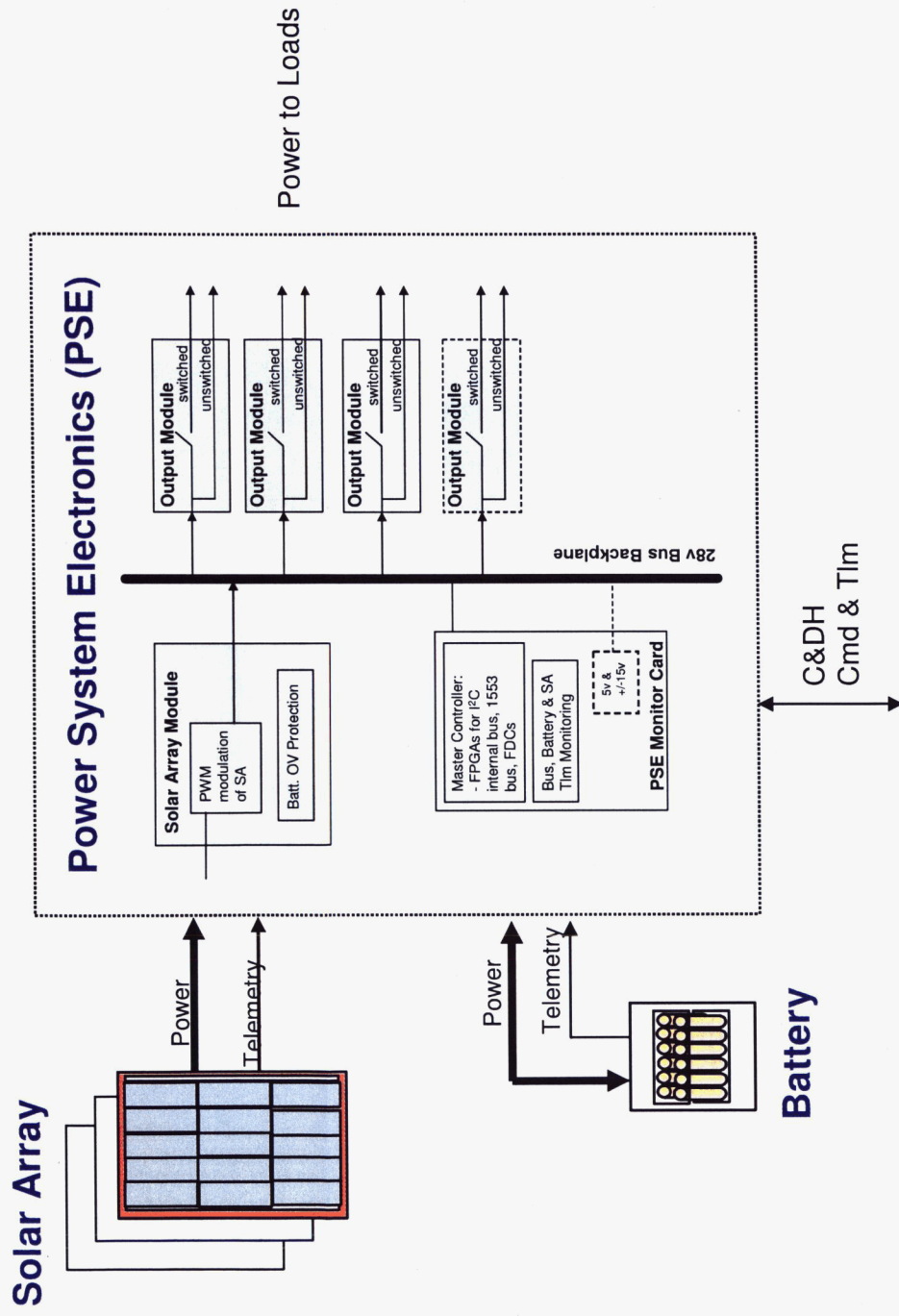
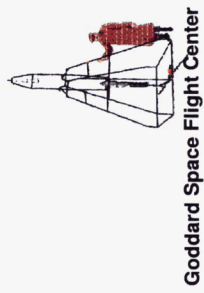


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- PSE
  - DET architecture with battery connected directly to the electrical bus
  - Provides unregulated bus between 22-35V
  - Single string
  - I<sup>2</sup> Serial Bus Interface
  - FPGA as Master Controller
  - Battery Charge Regulation is a combination of Hardware voltage and current control loops
    - Provide up to 30A max battery charge current
  - Power Distribution
    - Provide switched or un-switched power feeds to all spacecraft loads
    - Provide over-current protection on all feeds
  - Autonomous Fault Detection and Correction
    - Load shed
    - Battery Over-temp protection
    - Battery cell voltage protection

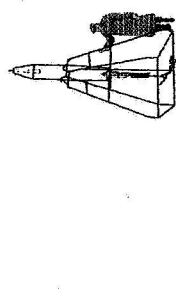


# LRO EPS Block Diagram

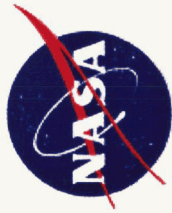




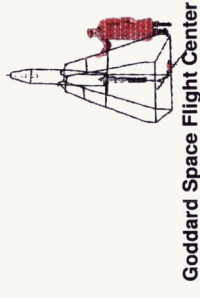
# History



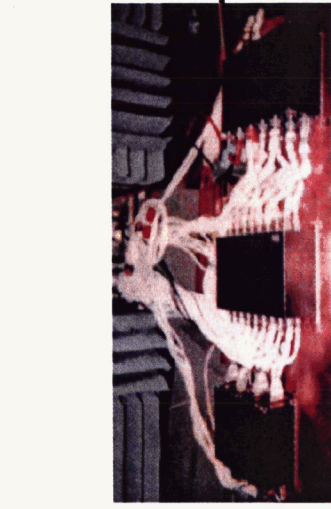
- XTE
  - Direct Energy Transfer System (DET)
  - Multiple boxes, harness for solar array interface, battery interface, data interface
- MAP / EO-1 Power System Electronics
  - DET System – Battery connected directly to bus
  - Solar array shunt with PWM Boost Converter for fine control
  - Modular Design:
    - SA Module, Distribution Module, Battery Module, Controller Module
- SDO Mission
  - Higher Power
  - Redundancy / Single Fault Tolerant
  - Longer Mission Life
  - New Deployment Function
  - PSE uses cPCI bus interface
- LRO Mission
  - I<sup>2</sup>C standard serial bus
  - FPGA implemented microcontroller, hardware compensation



# Advancement of PSE

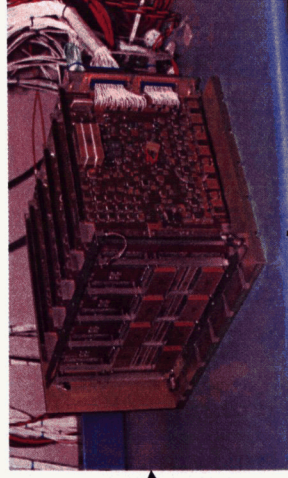


Optimizing the modular design: increase power to mass ratio



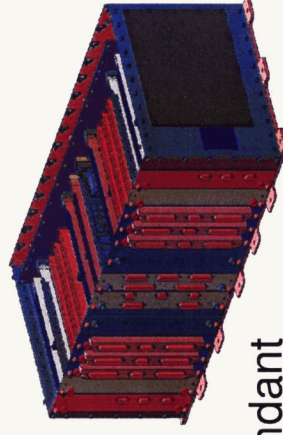
## XTE Redundant

- 40.4 Watt/kg
- 0.03 Watt/cm<sup>3</sup>
- 1995



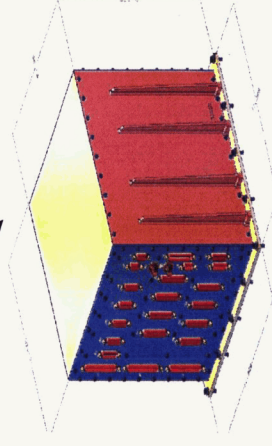
## MAP/EO-1

- 70.7 Watt/kg
- 0.04 Watt/cm<sup>3</sup>
- 2001



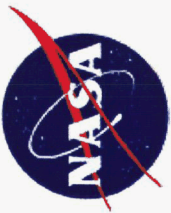
## SDO Redundant

- 65.6 Watt/kg
- 0.05 Watt/cm<sup>3</sup>
- 2008

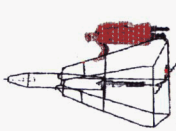


## LRO Single String

- 123.7 Watt/kg
- 0.13 Watt/cm<sup>3</sup>
- 2008

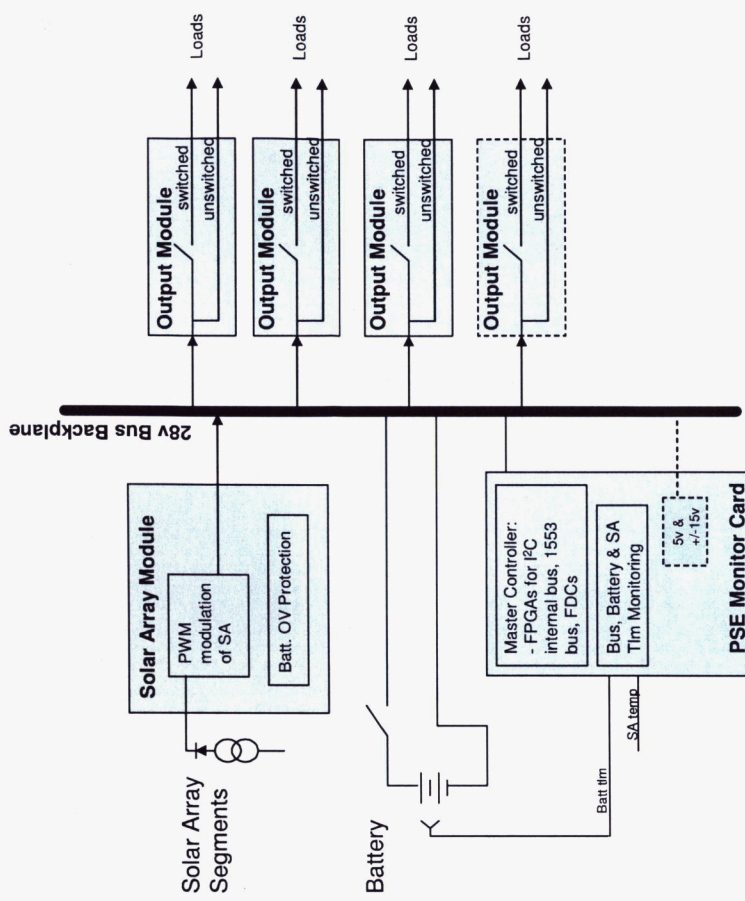


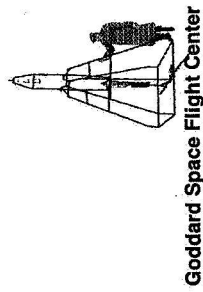
# PSE – Modular Design



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- Provide flexibility for spacecraft load configurations
- Increase power to mass ratio
- Eliminate internal harness
- Decrease architectural complexity
- Incorporate autonomous functions, i.e. FDCs
- Increase flexibility for battery management





# Summary

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- Modular PSE provides flexibility, robustness and reliability
  - Continue working to improve power to mass ratio
  - Increasing affordability due to reduced technology development and hardware costs
- Lithium Ion Batteries
  - On-going testing of Li-Ion batteries for current in-house missions (completed 2+ years of real-time LEO and GEO cycles)
  - Incorporation of lessons learned to Exploration Missions
- Solar Array
  - Incorporation of higher efficiency cells into future missions
  - Program to Reduce Risk
  - Incorporation of Hot Flash Test in Requirements